NACA TN No. 1885

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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DESIGN AND CALIBRATION OF A TOTAL-TEMPERATURE PROBE FOR USE AT SUPERSONIC SPEEDS

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Washington
May 1949

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SUMMARY

The results of a study of the factors involved in the design of total-temperature probes for use at supersonic speeds have been applied to the design of an instrument to be used in wind-tunnel calibrations. Tests of the probe showed that the calibration factor required in the conversion of the measured temperature to the true total temperature was 0.992 ± 0.002 for a range of Mach numbers between 1.36 and 2.01. The calibration factor was insensitive to angles of inclination relative to the air stream up to 3° for the range of Mach numbers from 1.36 to 2.01, and up to 9° at a Mach number of 1.50. These values of angle of inclination and Mach number were the limits of the range of test conditions for the investigation.

INTRODUCTION

In order to perform heat-transfer experiments in a supersonic wind tunnel at the Ames Laboratory, it was necessary to determine the total-temperature distribution in the wind-tunnel test section. A survey of the literature on the design of total-temperature probes for use at supersonic speeds indicated that there was not sufficient information available to design a suitable instrument. A discussion of the factors involved in the design of total-temperature probes intended for use at subsonic speeds can be found in a report by Hottel and Kalitinsky (reference 1). This report contains results from tests at subsonic velocities of bare-wire thermocouples and small shielded probes. Results of other investigations of the total temperature in high-speed air flows are to be found in references 2, 3, 4, 5, 6, and 7. These papers include results from the tests of both shielded and unshielded total-temperature probes in subsonic air flows.

It is the purpose of this report to present a discussion of some of the factors to be considered in the design of a total-temperature probe for use at supersonic velocities and to present data obtained with a specific design that was tested in the Ames 8- by 8-inch supersonic wind tunnel through the Mach number range from 1.36 to 2.01. Although the total-temperature probe for which test results are presented in this report was designed and tested for use in wind tunnels, it is believed that the instrument can be used to good advantage in other applications.

DESIGN CONSIDERATIONS

The design of a total-temperature probe to give a satisfactory calibration constant for a certain range of test conditions is largely a qualitative process involving the selection of materials and dimensions which will fit the test conditions. Previous investigations have shown certain design features to be desirable. Because of their small size and adaptability to precision measurement, thermocouples are generally used as the sensing elements in total-temperature measuring instruments. Franz (reference 2) found that it was necessary to place a flow-directing shield around the sensing element of a probe in order to make it insensitive to small angles of inclination relative to the stream direction. Wimmer (reference 4) found that good temperature recovery could be obtained just above the surface in the stagnation region of a sphere. These three features, the thermocouple temperature-sensing element, the flow-directing shield, and the position of the temperature-sensing element near the surface of a sphere form the basis of design for the type of probe considered in this report. (See figs. 1 and 2.)

The general energy equation for steady flow of a perfect gas with no gain or loss of energy is given by the expression

$$T_o = T_s + \frac{V^2}{2gJc_p} \qquad (1)$$

where

T_o total temperature, $^{\circ}F$ absolute

T_s static temperature which would be indicated by a thermometer moving at the same velocity as the stream, $^{\circ}F$ absolute

V velocity of the stream, feet per second

g acceleration due to gravity, feet per second squared

J mechanical equivalent of heat, 778 foot-pounds per Btu

c_p specific heat at constant pressure, Btu per pound, $^{\circ}F$

Thus it can be seen that the total temperature of a moving stream is the sum of two factors, the static temperature and a factor that is a measure of the kinetic energy.

If it were possible to bring the stream to rest adiabatically at a thermocouple, the kinetic energy of the stream would be completely recovered and the thermocouple would indicate the total temperature. With an actual temperature probe, however, it is impossible to achieve the condition of adiabatic internal air flow. As the temperature of the air sample, in its path into and through the probe, increases above the static temperature of the free air stream, there is a simultaneous loss of heat from the air sample and the thermocouple due to both radiation to surrounding surfaces and conduction through the shield and along the thermocouple wires. Therefore, the thermocouple indicates some temperature T that lies between the static and total temperature of the air stream. The equation relating the temperatures in this case can be given by the expression

$$T = T_s + C \frac{V^2}{2gJc_p} \tag{2}$$

where C, the calibration factor, is a measure of the effectiveness of the probe in recovering the kinetic energy of the air stream at the thermocouple.

Combining equations (1) and (2) results in an expression for the calibration factor

$$C = \frac{T-T_s}{T_o-T_s} \tag{3}$$

If the value of C is known as a function of Mach number from a calibration for a given instrument, it is only necessary to know the free-stream Mach number and the indicated total temperature T to determine the true total temperature. In general, it is necessary

to have the minimum possible heat loss from the air sample and thermocouple junction in order to have an accurate instrument because, if the heat loss is small, changes in heat loss due to changing test conditions will be negligible. The result is a value of the calibration factor C which is essentially constant and near unity over a range of test conditions. A constant value of calibration factor is advantageous in that it simplifies the reduction of test data.

The problem in a shielded probe, then, consists of getting an air sample to the thermocouple junction with the minimum heat loss through the shield and arranging the thermocouple junction within the probe so that it will receive the maximum amount of heat from the air sample and lose the minimum amount of heat through radiation and conduction. The conduction losses from the air inside the shield are probably the major losses. These losses are due to the difference between the recovery temperature of the slow-speed internal air flow and the recovery temperature attained by the air on the outside surface of the shield. This difference is large at supersonic speeds. In addition, losses by radiant-heat transfer occur from the surface of the probe to surroundings which are at temperatures different from that of the probe surface. These conduction and radiation losses can be minimized by the proper selection of the shield shape, materials, and ratio of vent area to entrance area.

There are two general possibilities for the shape of the shield for a probe to be used in supersonic air flows. It can either be long and thin with an attached bow wave or it can be short and blunt with a detached bow wave. (The position of the bow wave depends to a large extent on the ratio of vent to entrance area.) The long shield has the disadvantages of having considerably more surface area than the short one and of having a thin, sharp edge on the entrance lip. This thin edge, with the thin internal and external boundary layers, will allow considerable heat loss in relation to the short shield with its thick, blunt lip and relatively thick boundary layers. In the case of the short probe there is a possibility of heat loss from the air sample by conduction in the region between the bow wave and the entrance, but this loss is believed to be relatively small. Also, the effect of angle of inclination on the calibration factor could be expected to be less with a blunt shield than with a long, thin shield because there is less chance for internal flow separation due to either internal shock-boundary-layer interaction, or large adverse pressure gradients at the probe entrance.

The air sample enters the blunt shield at a subsonic velocity behind the detached bow wave and is generally diffused to a still lower velocity in an expanding duct within the shield, although a cylindrical duct through the shield should be adequate. Hottel and Kalitinsky (reference 1) have shown that it is necessary to have a high Reynolds number, based on entrance diameter and flow conditions, so that the internal boundary layer will not fill the diffusor. If the internal boundary-layer thickness approaches the duct radius, the effect of conduction through the shield will be felt by the filament of the air sample which passes over the thermocouple junction, and the calibration factor will be decreased. This would indicate the desirability of large probes and of appreciable internal velocities. If a diffusor is to be used, care must be taken in the selection of the diffusor angle and the ratio of vent to entrance area so as to prevent flow separation from the diffusor walls. If separation occurs, the calibration factor will be reduced because of the mixing of the relatively cool wall boundary layer with the central filament of the air sample.

The heat loss from the air sample by conduction through the shield can be minimized by using a material of low thermal conductivity and by providing an internal air gap between the inner and outer walls. This air gap could be evacuated to further reduce the thermal conductivity. The flow-directing shield can also become an effective radiation shield if its surfaces are coated with a material of low emissivity. To be practical, this material must be hard and tarnish resistant.

After the velocity of the air sample is reduced in the diffusor with the least possible heat loss through the walls, the temperature of the air is measured with a thermocouple. According to the results of reference 4, the thermocouple junction should be located just off the surface of the hemisphere within the shield. At this location, the junction is in a region of very low, but not zero, airspeed and is effectively insulated from the hemisphere by the short, fine thermocouple wires. This position provides a greater heat transfer to the thermocouple junction than would be obtained if the junction were located at the zero-velocity point on the surface of the hemisphere. In a shielded total-temperature probe the ratio of vent area to entrance area determines the air velocity and boundary-layer thickness over the hemisphere which in turn determine the optimum location of the thermocouple junction. The final step in the development of a probe is, therefore, almost invariably the experimental determination of the optimum ratio of vent area to entrance area.

APPARATUS AND TESTS

Figures 1 and 2 show the details of the total-temperature probe designed according to the foregoing considerations. The probe was made of lucite because of the low value of thermal conductivity of this material and the ease with which it could be fabricated. The probe was machined in three pieces and cemented together after an iron-constantan thermocouple of 30-gage wire had been fixed in place. The thermocouple junction was located 0.050 inch ahead of the stagnation point on a hemisphere, which was the position found to be most desirable by Wimmer (reference 4). An air gap was provided around the diffusor to act as additional insulation against heat loss from the sample air. The lucite nose shield was considered adequate as a radiation shield because of the moderate temperatures, 50° to 110° Fahrenheit, at which the probe would be required to operate.

The probe was calibrated in the Ames 8- by 8-inch supersonic wind tunnel, a continuous, open-circuit wind tunnel. The nozzle of this wind tunnel is adjustable and the Mach number can be varied between 1.2 and 2.1 while the tunnel is in operation. During the tests the total pressure was varied from about 23 pounds per square inch absolute at a Mach number of 1.36 to 39 pounds per square inch absolute at a Mach number of 2.01. The total temperature during the tests was approximately 60° Fahrenheit. The absolute humidity of the air in the wind tunnel was held below 0.0001 pound of water per pound of dry air by removing the moisture in silica-gel dryers. The total temperature of the tunnel, against which the probe was calibrated, was measured by nine iron-constantan thermocouples located on a turbulence-reducing screen in the settling chamber upstream from the nozzle. The temperatures indicated by these thermocouples and the thermocouple in the total-temperature probe were read on an indicating pyrometer-potentiometer equipped with an automatic compensation for the cold junction. The pyrometer-potentiometer was sensitive to an unbalance of 0.1° Fahrenheit which is equivalent to a maximum variation of ± 0.002 in the value of the calibration factor.

The probe was supported from the rear so that the probe inlet was located at the center of the air stream in the test section of the wind tunnel. The angle of inclination between the axis of the probe and the stream direction could be varied from 0° to 9° in such a manner that the probe inlet remained approximately in the center of the test section.

During the tests the nine thermocouples in the settling chamber showed, in general, a variation of less than 1° Fahrenheit. The

greatest variation occurred between the five thermocouples located near the center of the settling chamber and the remaining four thermocouples near the settling-chamber wall. The four thermocouples near the wall consistently indicated slightly higher temperatures than those near the center of the settling chamber. This temperature difference was probably due to the heat transfer through the settling-chamber wall from the room to the cooler tunnel air. The five thermocouples near the center of the settling chamber generally showed a variation of less than 0.2° Fahrenheit and the average of their readings was used in the calculation of the calibration factor.

The use of the five center thermocouples to measure the true stagnation temperature was justified by tests of a similar installation in the Ames 1- by 3-foot supersonic wind tunnel No. 1 in which the total temperature could be controlled. The test showed that with the total temperature of the tunnel equal to room temperature all the thermocouples in the settling chamber read the same and resulted in a certain calibration factor for the probe which was mounted in the center of the test section. When the total temperature of the tunnel was increased to 30° Fahrenheit above room temperature, the thermocouples near the settling-chamber wall were influenced by the heat transfer through the wall. However, by using the thermocouples near the center of the settling chamber the same calibration factor was obtained for the probe as was determined with all the settling-chamber thermocouples reading alike when the total temperature of the tunnel was equal to room temperature.

The test program was conducted in three phases. In the first phase, the Mach number was held constant at 1.50 and the size of the vent holes was varied to determine the optimum ratio of vent area to entrance area. Two vent holes of 1/16-inch diameter were drilled initially and the diameter was increased by 1/32-inch increments to 1/8 inch after which two additional 1/16-inch diameter holes were drilled and the diameter of the new holes was increased by 1/32-inch increments. In the second phase of the test program the free-stream Mach number was varied and its effect on the calibration factor was measured. Data were obtained at Mach numbers of 1.36, 1.50, 1.69, 1.90, and 2.01 using the optimum ratio of vent area to entrance area obtained at a Mach number of 1.50. The third phase of the test program consisted of varying the angle of inclination of the probe and determining the effect on the calibration factor. Data were obtained at Mach numbers of 1.36, 1.50, 1.69, 1.90, and 2.01 for an angle of inclination of 3° and at a Mach number of 1.50 for angles of inclination of 6° and 9°.

RESULTS

The results of the calibration of the total-temperature probe are shown in figures 3, 4, and 5. The effect on the calibration factor of varying the ratio of vent area to entrance area is shown in figure 3. It can be seen from the data in this figure that the optimum ratio of vent area to entrance area is between the values of 0.500 and 0.625 for a Mach number of 1.50.

Figure 4 shows the results of the calibration of the total-temperature probe with Mach number for a ratio of vent area to entrance area of 0.625. It is apparent that within the limits of accuracy of the test (±0.002) the calibration factor is constant at a value of 0.992 through the Mach number range from 1.36 to 2.01.

It can be seen from figure 5 that the total-temperature probe is insensitive to angles of inclination up to 9° at a Mach number of 1.50. Data also obtained at an angle of inclination of 3° show the calibration factor to be constant at all Mach numbers investigated.

CONCLUSIONS

The results obtained from tests at supersonic velocities of a total-temperature probe design based on the available experimental data and a qualitative study of the factors involved show the following characteristics:

1. The calibration factor required in the conversion of the measured temperature to the true total temperature was 0.992 ± 0.002 at angles of inclination relative to the air stream of 0° and 3° for a range of Mach numbers between 1.36 and 2.01.

2. The calibration factor was insensitive to angles of inclination relative to the air stream up to 9° at a Mach number of 1.50.

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REFERENCES

1. Hottel, H. C., and Kalitinsky, A.: Temperature Measurements in High-Velocity Air Streams. Jour. of App. Mech., Mar. 1945, pp. A 25 – A 32.

2. Franz, A.: Pressure and Temperature Measurement in Supercharger Investigations. NACA TM No. 953, 1940.

3. Eckert, E.: Temperature Recording in High-Speed Gases. NACA TM No. 983, 1941.

4. Wimmer, W.: Stagnation Temperature Recording. NACA TM No. 967, 1941.

5. Lindsey, W. F.: Calibration of Three Temperature Probes and a Pressure Probe at High Speeds. NACA ARR, 1942.

6. King, W. J.: Measurement of High Temperatures in High Velocity Gas Streams. General Electric Company, 1942.

7. Markowski, S. J., and Moffatt, E. M.: Instrumentation for Development of Aircraft Powerplant Components Involving Fluid Flow. S.A.E. Quarterly Transactions, vol. 2, no. 1, Jan. 1948, pp. 106–111.

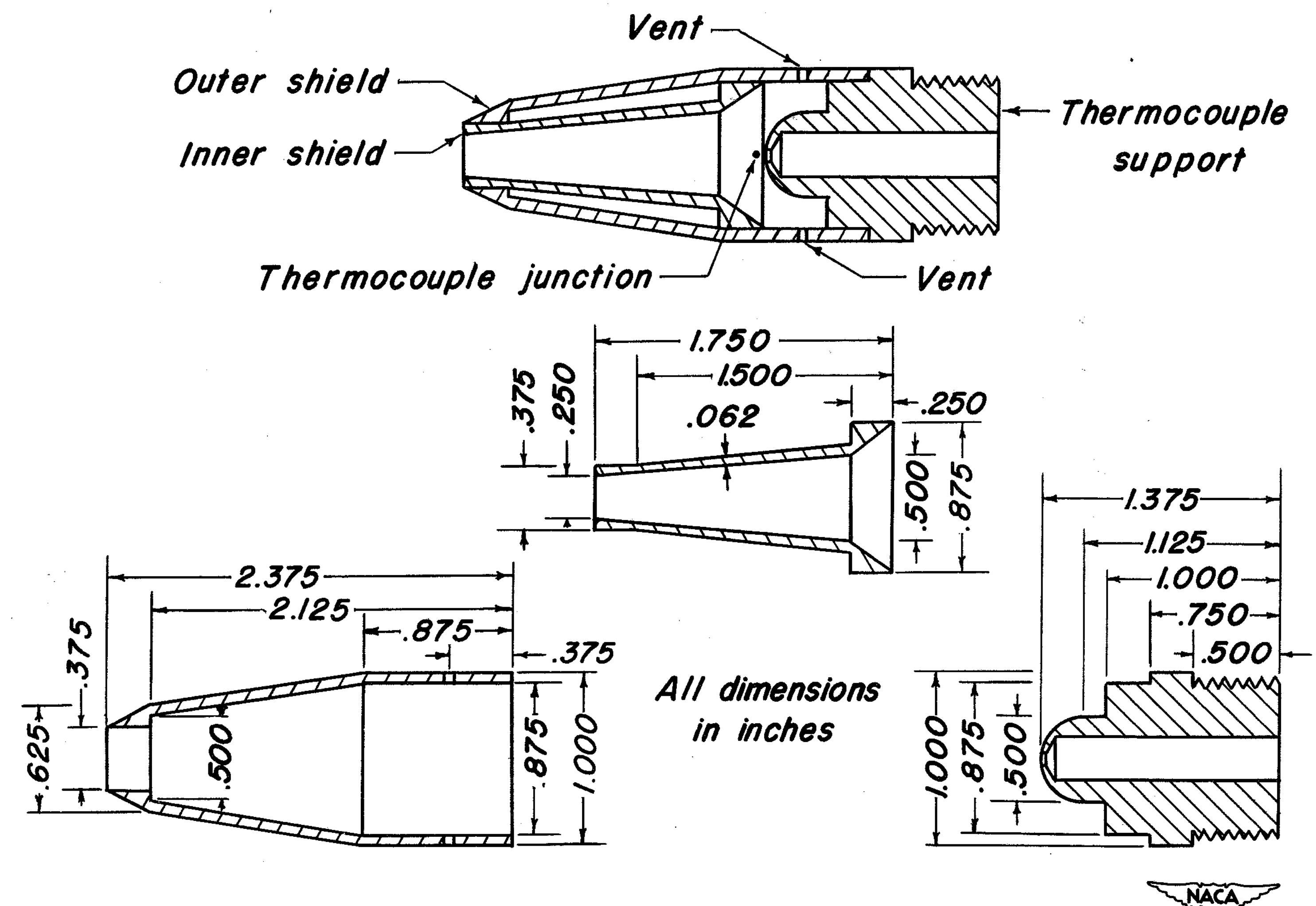

Figure 1. —Total—temperature probe dimensions.

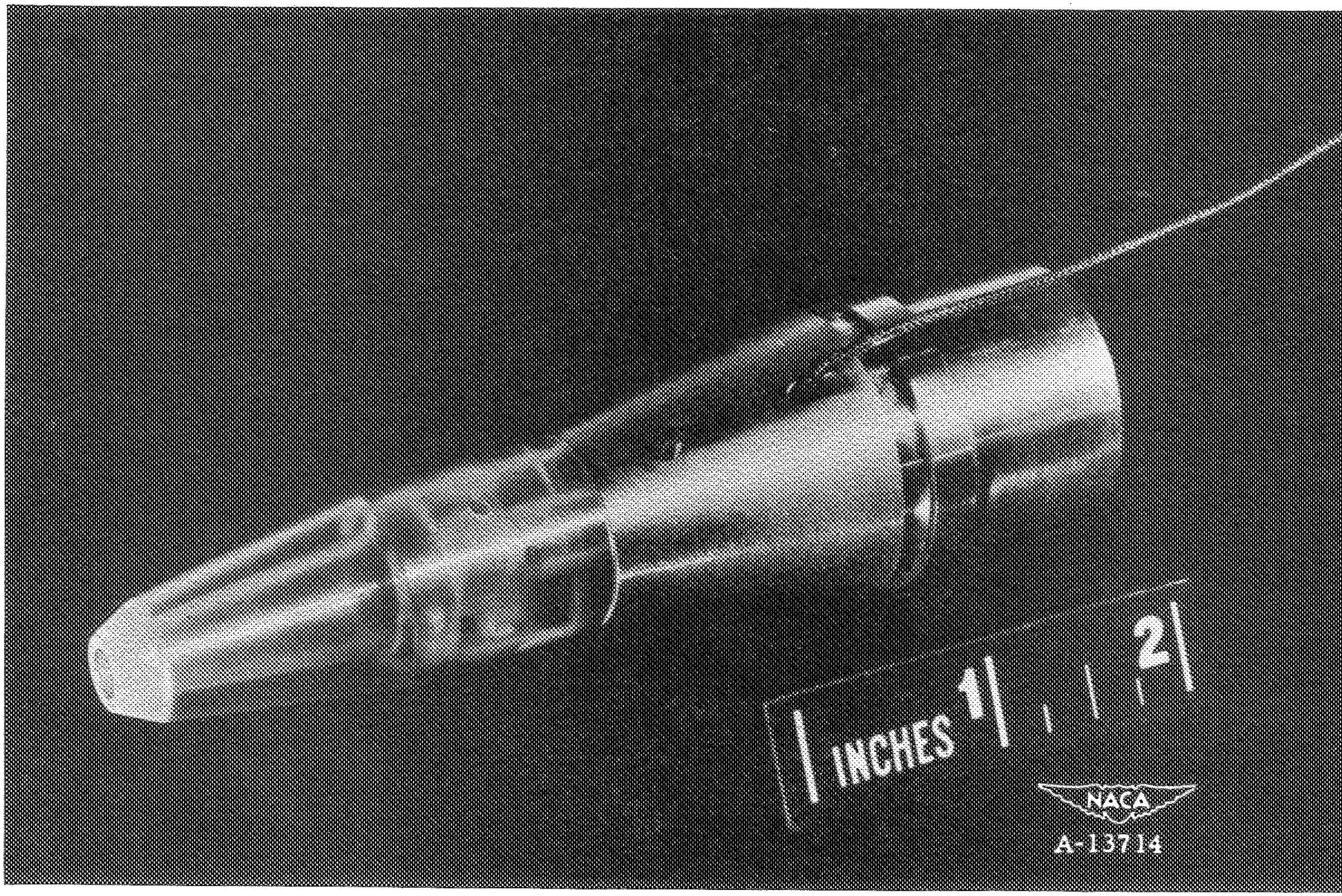

Figure 2.— Total-temperature probe and support.

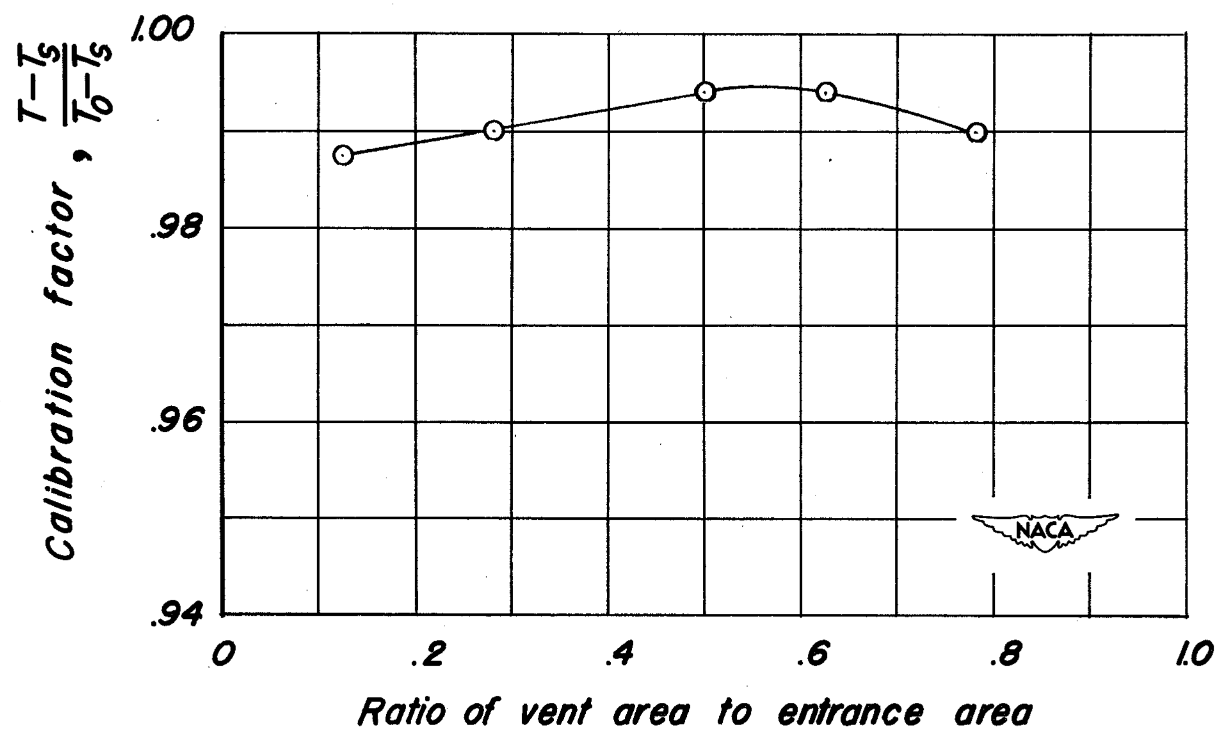

Figure 3.— Variation of calibration factor with ratio of vent area to entrance area at a Mach number of 1.50.

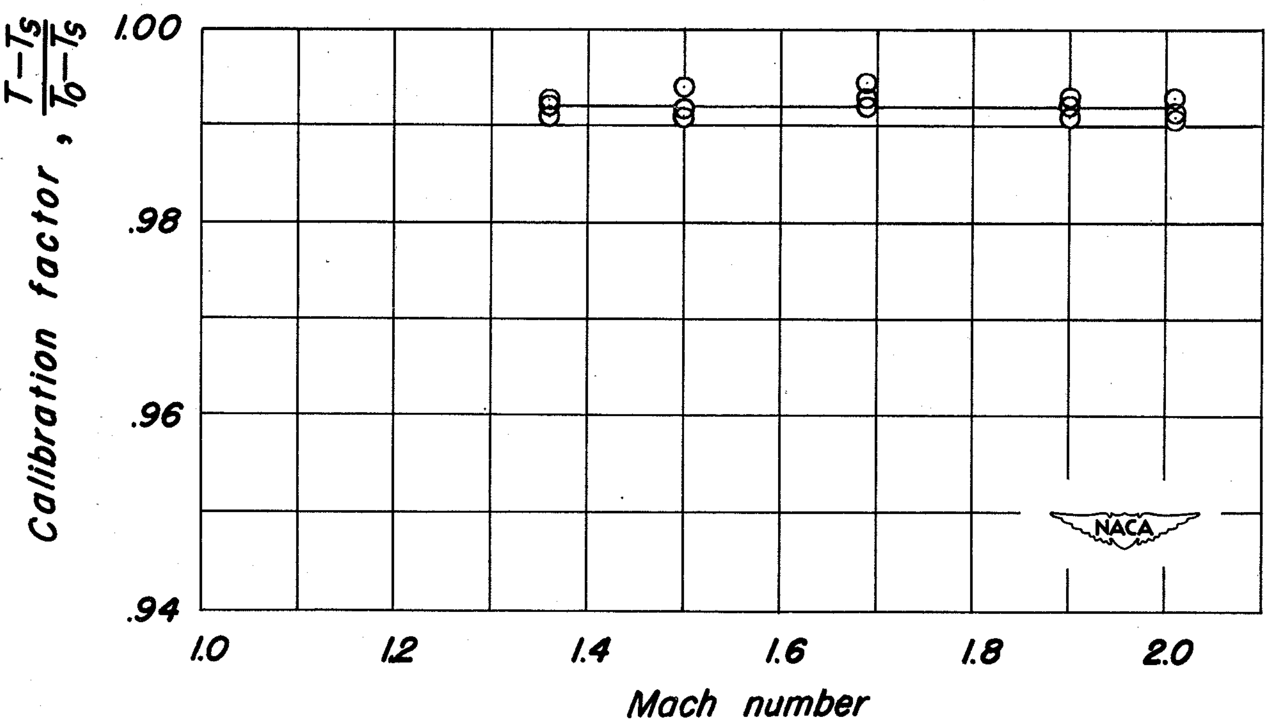

Figure 4.— Variation of calibration factor with Mach number for a ratio of vent area to entrance area of 0.625.

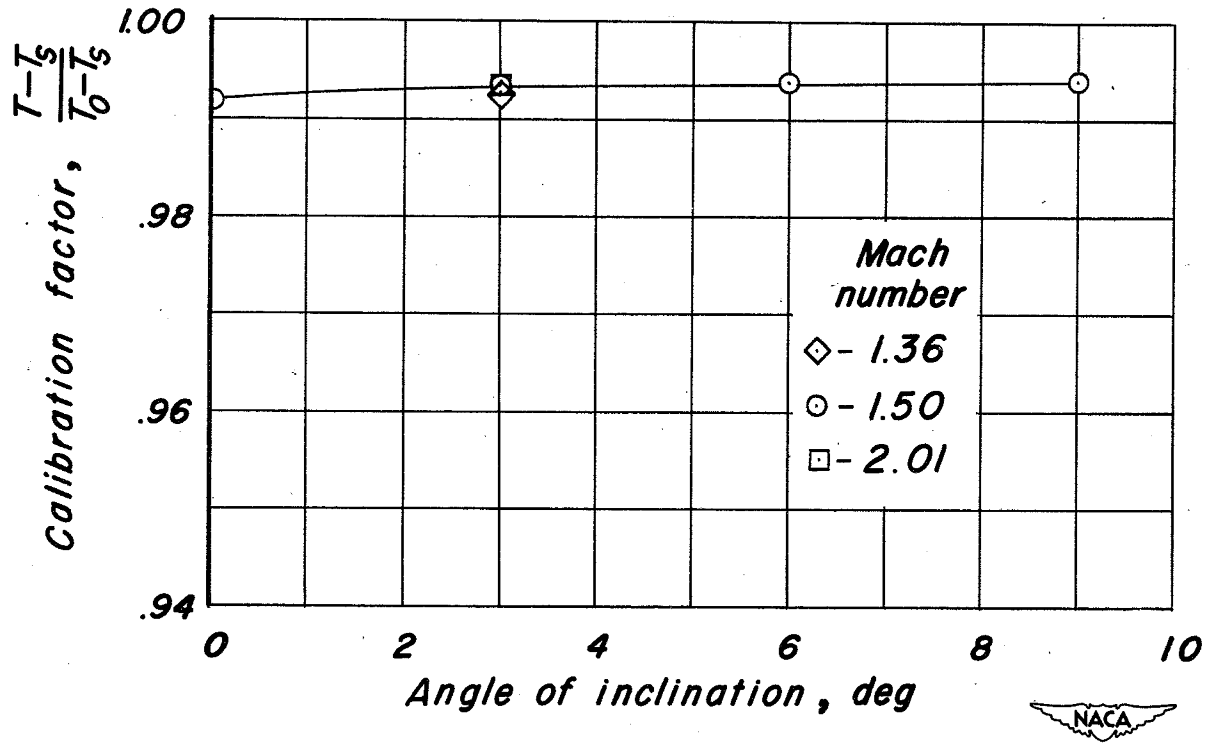

Figure 5.— Variation of calibration factor with angle of inclination for a ratio of vent area to entrance area of 0.625.